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Compressor Gas Turbine Matching with Reference to Bio Fuel Firing Systems

Gopalakrishnan G, Provost
Dr. M. G. R. Educational & Research Institute, Chennai 60095, India

There have been in recent time's research being extended to the utilization of gas turbines adopting bio fuels. The main idea is that a large number of turbines after aviation /defense uses is normally confined to the storage yards and utilized for reclamation of auxiliary units or metals. Not to be differentiated from the normal use of JP4 or kerosene as fuels, a simplified gas turbine matching parametric relationship has been established. Considering a normal Brayton cycle, relationships have been arrived at for various parametric variables. This is a conceptual paper.

Keywords: *Gas turbines for bio fuels, bio fuelling gas turbines, renewable energy turbines*

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I. Introduction

Considering a normal Brayton Cycle, the general cycle parameters have been derived. Of late, there have been several cases where non-fossil fuels have been utilized to help generate power employing gas turbine. The system consists of a compressor, a combustion chamber a turbine and a nozzle, the gas turbine operating on the same 'axis'; the compressor and the turbine are on the same single shaft, as shown in the following figure
In Fig.1, a general Brayton cycle gas turbine T-S diagram has been presented. In the actual compressor turbine matching, it is assumed that a nozzle and a diffuser are attached to the system being discussed.

1. Compressor-Turbine Matching in a Gas Turbine

The System

The system consists of a compressor, a combustion chamber a turbine and a nozzle, the gas turbine operating on the same 'axis'; the compressor and the turbine are on the same single shaft, as shown in the following figure. As seen from the T- S diagram in Fig. 1, air is drawn into the diffuser ahead of the compressor,0, 1 – 2 (air being drawn from the atmosphere does involve a drop in the temperature from 0-1, not shown). The air then passes through the compression process in the Compressor, 2 - 3. The compressed air then gets through the Combustor, combustion of fuel air takes place 3 - 4, and then expanded through the Turbine 4 - 5 and then again through the Nozzle 5 – 6. (Fig.2). This is the process of a turbofan jet engine.

A more detailed review of gas turbine performances and aerodynamics of turbomachines could be obtained from (Horlock 1966), (Gopalakrishnan 2019), (Cohen, H., Saravanamuttoo, H. I. H., et al., 2001)and a host of others.

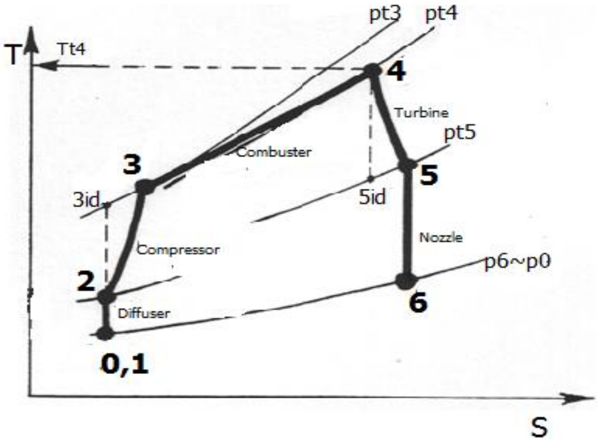

Figure 1 T-S diagram for the System – Diffuser, Compressor, Combustor, Turbine and the Nozzle

2. Assumptions

The following assumptions hold:

i. Processes 0-2 (Diffuser) and 5-6 (Nozzle) assumed isentropic
ii. Not much of thermodynamic differences between Stations 0 and 1
iii. The Compressor and the turbine are on the same shaft/single spool
iv. Fuel air ratio (f) ~ Mass of fuel burnt ($\dot{m}_f$)/ Mass of air delivered by the compressor ($\dot{m}_a$)
v. All variables of the compressor, turbine and the air, fuel and other inputs are known
vi. Nozzle exhausts into the atmosphere, i.e., p_6=p_0

3. Basic Parameters

Compressor Inlet Air flow:

In order to assess the inflow through the compressor inlet, a bell mouth may be attached (DIN/ISO standard nozzle would be preferred). Assuming that there is slight depression between the atmospheric state at inlet to the nozzle, p_0, and at inlet to the compressor p_1, it follows that $p_1 = p_0 - \partial p$ where '∂p' is the differential pressure.

The Mach number at inlet becomes,

$$M_{inlet} = \sqrt{\frac{2}{\gamma_a - 1}\left[\left(\frac{cc.p_0}{p_1}\right)^{\frac{\gamma_a - 1}{\gamma_a}} - 1\right]}$$

$$T_1 = T_0 \,/\, (1+\frac{\gamma_a - 1}{2}.\, M^2)$$

$$\rho_1 = p_1/(R_a.\, T_1)$$

$$V = M.\, a = M.\sqrt{(\gamma_a\, .R_a\, .T_0)}$$

$$A = \frac{\pi}{4} D^2$$

$$M_{air\ flow} = \rho_1\, .\, V.\, A$$

For a bell mouth at inlet the coefficient varies between 0.9…0.95

Since,

$$p_1 = p_0 - \partial p$$

Mach no. at inlet is now,

$M_{inlet} = \sqrt{\frac{2}{\gamma_a - 1}\left[\left(\frac{p_0}{p_1}\right)^{\frac{\gamma_a - 1}{\gamma_a}} - 1\right]}$, the coefficient of contraction taken into account.

The following temperature and pressure ratios apply:

$$\tau_d = \frac{T_{t2}}{T_0} = \{1+\frac{\gamma-1}{2} M_0^2\};\ \tau_{comp} = \frac{T_{t3}}{T_{t2}};\ \ \tau_{cc} = \frac{T_{t4}}{T_{t3}};\ \ \tau_{turb} = \frac{T_{t5}}{T_{t4}} < 1.0\,; \tag{1}$$

$$\pi_d = \frac{p_{t2}}{p_0} = \{1 + \frac{\gamma-1}{2} M_0^2\}^{\frac{\gamma}{\gamma-1}};\ \pi_{comp} = \frac{p_{t3}}{p_{t2}};\ \pi_{cc} = \frac{p_{t4}}{p_{t3}};\ \pi_{turb} = \frac{p_{t5}}{p_{t4}} < 1.0; \text{ and } p_{t6} \cong p_{t0}$$

Where, 'τ' denotes the temperature ratios while 'π' denotes the pressure ratios, and the subscripts 'diff', 'comp', 'cc' and 'turb' denote the diffuser, compressor, combustion chamber and the turbine respectively. As it has been indicated earlier that the turbojet system being matched is of a single spool type, implies that the compressor and the turbine run at the same speed, i.e.,

$$N_{turb} = N_{comp} = N \text{ (revs. / mt.)} \tag{2}$$

For mass continuity of the flow through the system, the air from the compressor along with the fuel inject and combusted would flow through the turbine, i.e.,

$$\dot{m}_{turb} = \dot{m}_{comp} + \dot{m}_{fuel}$$

$$= \dot{m}_{comp}\,\{1 + \frac{\dot{m}_{fuel}}{\dot{m}_{comp}}\} \text{ alternately,}$$

$$\dot{m}_4 = \dot{m}_2\ \{1+ f\} \text{ where, 'f' is the fuel/air ratio.} \tag{3}$$

The power produced by the turbine of the turbojet, goes to drive the compressor and the fan as well. Hence the power of the turbine should be balanced by the power consumed by the compressor and the fan together and of course the frictional components, considered negligible.

$$\eta_m\, \dot{m}_{turb} c_{p_{turb}} (T_{t4} - T_{t5}) = \dot{m}_{comp}\ c_{p_{comp}}\ (T_{t3} - T_{t2})$$

$$\eta_m\, (\dot{m}_{c0mp} + \dot{m}_{fuel})\ c_{p_{turb}}\ (T_{t4} - T_{t5}) = \dot{m}_{comp}\ c_{p_{comp}}\ (T_{t3} - T_{t2})$$

$$\eta_m\, (1+f)\ c_{p_{turb}}\ T_{t4}\ (1 - \frac{T_{t5}}{T_{t4}}) = c_{p_{comp}}\ T_{t2}\ (\frac{T_{t3}}{T_{t2}} - 1)$$

$$\tau_{turb} = \{1 - \frac{1}{\eta_m\,(1+f)} . \frac{c_{p_{comp}}}{c_{p_{turb}}} . (\frac{T_{t2}}{T_{t4}}) . (\tau_{comp} - 1)\} \tag{4}$$

Here it must be taken into account, that,
$c_{p_{comp}} = (\frac{1.41}{1.41-1})$ x 287(J/kg. K) ; $c_{p_{turb}} = (\frac{1.33}{1.33-1})$ x 287 (J/kg. K)
From the equations, it is possible to fix the points on the compressor characteristics along with the points on the turbine characteristics. Since for all practical purposes the flight Mach number, M_0 and the ambient conditions T_0 and p_0 and the real physical mass flow rate $\dot{m}_{comp}$ are known, and also since the corrected mass flow rate $\dot{m}_{c2}$ the corrected speed N_{c2} can be calculated, the point on the compressor characteristic could now be located.

$$\theta_2 = \frac{T_{t2}}{288°K} ; \quad (5)$$

$$\delta_2 = \frac{p_{t2}}{1\ bar} ; \quad (6)$$

$\dot{m}_{comp\ c2} = \frac{\dot{m}\sqrt{\theta_2}}{\delta_2}$ … corrected mass flow rate, compressor (subscript 'c' stands for corrected)
$N_{comp\ c2} = \frac{N}{\sqrt{\theta_2}}$ … corrected speed, compressor (subscript 'c' stands for corrected) It is now possible to locate the compressor operational point on the compressor characteristics. With the calculated $\frac{\dot{m}\sqrt{\theta_2}}{\delta_2}$ for the calculated speed $\frac{N}{\sqrt{\theta_2}}$ from the compressor characteristic, the pressure ratio at which the compressor operates $\pi_{comp} = \frac{p_{t3}}{p_{t2}}$ can now be established.
The turbine inlet temperature T_{t4} is the system's cycle maximum temperature; the fuel air mixture and the mechanical efficiency η_m are known data. The total to total temperature for the turbine stage could be calculated i.e., $\tau_{turb} = \frac{T_{t5}}{T_{t4}}$ .Since through the compressor characteristic, we have now obtained, the pressure ratio $\pi_{comp} = \frac{p_{t3}}{p_{t2}}$ and the compressor efficiency at the point of operation η_{comp} it is now possible to arrive at the following:

$\eta_{comp} = \frac{\pi_c^{\frac{\gamma-1}{\gamma}}}{\tau_c - 1}$ (τ_{comp} can now be computed)

$\frac{c_{p_{comp}}}{c_{p_{turb}}} = \{(\frac{\gamma_{comp}}{\gamma_{comp-1}} *R) / (\frac{\gamma_{turb}}{\gamma_{turb-1}} *R) \approx 0.8684$

τ_{turb} can now be computed from Eq(4),

$\tau_{turb} = \{1 - \frac{1}{\eta_m\ (1+f)} . \frac{c_{p_{comp}}}{c_{p_{turb}}} . (\frac{T_{t2}}{T_{t4}}) . (\tau_{comp} - 1)\}$

Since T_{t2} and T_{t4} are now known the turbine corrected speed could be assessed;

$$\frac{N}{\sqrt{\theta_4}} = \{\frac{N}{\sqrt{\theta_2}} * (\frac{T_{t2}}{T_{t4}})\} \quad (7)$$

Using a trial and error method, it is now necessary to determine the turbine operational point in the turbine characteristics.
The turbine pressure ratio map is $\frac{p_{t4}}{p_{t5}} = \frac{1}{\pi_{turb}}$ vs = $\frac{\dot{m}\sqrt{\theta_4}}{\delta_4}$ Using this turbine characteristic map, continue calculating τ_{turb}
Noting the efficiency η_{turb} , it is now possible to obtain τ_{turb} .

$$\tau_{turb} = 1 - \eta_{turb}\ (1 - \pi_{turb}^{\frac{\gamma-1}{\gamma}}) \quad (8)$$

The process is now repeated for another value of speed across the characteristic till the values match.
The corrected mass flow rate on the turbine characteristic,

$\frac{\dot{m}_{4c}\sqrt{\theta_4}}{\delta_4} = \frac{\dot{m}_{4c}\sqrt{\frac{T_{t4}}{288°K}}}{\frac{p_{t4}}{1\ bar}}$ where $\dot{m}_4 = (1+f)\dot{m}_2$, from this p_{t4} can be evaluated;
accordingly the total pressure drop across the combustion chamber (p_{t3} - p_{t4}). Having calculated the total pressures and temperatures at the turbine inlet and exit and also the pressure p_{t5} (assumed equal to p_{t6} , as the nozzle flow is assumed to be isentropic) and T_{t5} assumed equal to T_{t6} the nozzle exit temperature, it is possible to now compute M_6. .

$$p_{t5} = p_{t6} = p_6 = p_0 = p_6\ [1 + \frac{\gamma-1}{2} M_6^2], \quad \text{where } p_0 \text{ is the ambient pressure} \quad (9)$$

$T_{t5} = T_{t6} = T_6\ [1 + \frac{\gamma-1}{2} M_6^2]$ in which M_6 is known from Eq(9).

5.1 Sonic Speed may now be calculated as α_6 ,

$$\alpha_6 = \sqrt{\gamma R T_6} \text{ , the nozzle exit velocity } V_6 = M_6 . \alpha_6 \quad (10)$$

5.2 Specific Thrust

$$\frac{F}{\dot{m}_2} = (1+f)V_6 - V_0 \text{ , where } V_0 = M_0 . \alpha_0 = M_0 \sqrt{\gamma R T_0} \quad (11)$$

5.3 Specific Fuel Consumption could be decided on the energy balance method;
Rate of thermal energy input + Heat produced by fuel combustion = Rate of thermal energy leaving the combustion chamber, and this could be represented as,

$$\dot{m}_2 \, c_{p_{comp}} \, T_{t3} + \dot{m}_f * h = (\dot{m}_2 + \dot{m}_f). \, c_{p_{turb}} \, . \, T_{t4} \quad (12)$$

Where, ($c_{p_{comp}}$ *and* $c_{p_{turb}}$ have $\gamma = 1.41$ and 1.33 respectively, while 'h' is the heating value of the fuel)

5.4 Fuel Air Ratio

$$f \equiv \frac{\dot{m}_f}{\dot{m}_2} = \frac{(c_{p_{turbo}} T_{t4} - c_{p_{comp}} T_{t3})}{(h - c_{p_{turbo}} T_{t4})} \quad (13)$$

5.5 Specific Fuel Consumption

$$\text{s.f.c.} = \frac{f * 10^6}{(\frac{F}{\dot{m}})} = \{\frac{\text{mg.fuel/sec}}{\text{N.thrust}}\} \quad (14)$$

II. Discussions

The above equations are equally valid for both JP4/Kerosene fired gas turbines as well as those for gas turbines being run on renewable stock like ground nut shells, paddy/wheat husks, bamboo chips, wood chips, etc being burnt in a fluidized bed heating system which goes to heat the air from the compressor before entering the turbine. However, the system is to be experimented before being sent into circuit.

A large amount of experimentation needs to be done . However some research is ongoing on the use of solar energy to heat the compressed air further after getting out of the compressor instead of using fossilized fuels in the cannular combustors of an normal gas turbine engine.

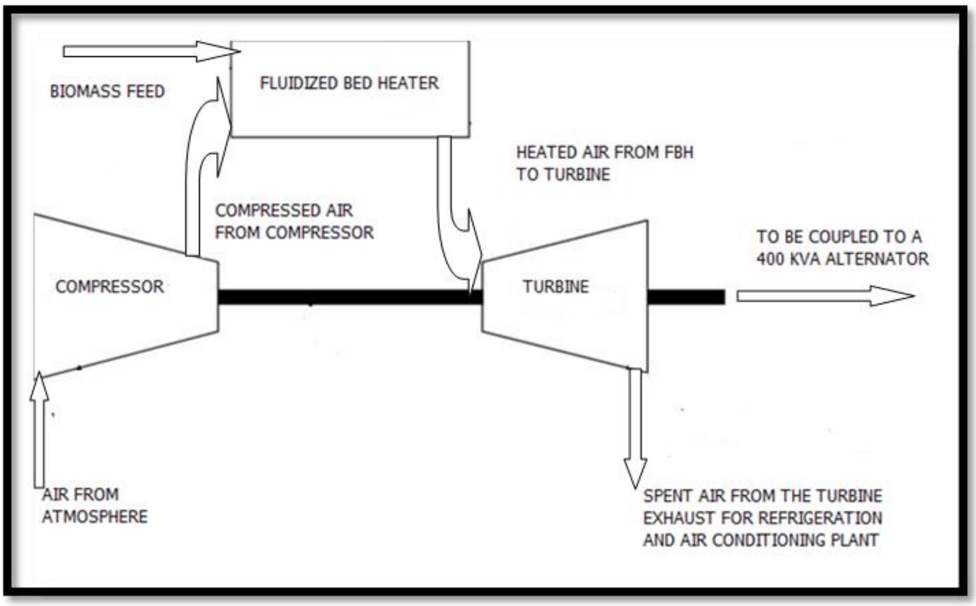

Figure 2 Flow Diagram - *Biomass Fuelled Aero Derivative Gas Turbine*

III. Conclusions

Normally, the existing gas turbine has to be modified to suit a bio-fuel that is intended to be used. The compressed air from the compressor needs to be brought out of the engine air frame and connected to the bio-fuel/solar firing/heating system. From the bio-fuel burner / solar heating, the air then expands in the turbine; thereby transferring power to the turbine mounted on the turbo shaft extension. The heat depleted air from the turbine exhaust could also depending on the turbine exhaust temperatures, be used for an external refrigeration cycle. With the above deductions and details worked out, it is possible to work out the Compressor-Turbine matching conditions of a turbojet engine, for any type of bio-fuel being used.

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References

[1]. Horlock, J. H., 1966, Axial Flow Turbines, Fluid Mechanics and Thermodynamics, Butterworths, UK
[2]. Cohen, H., Saravanamuttoo, H. I. H., et al., 2001, Gas Turbine Theory, Pearson Education, UK
[3]. Gopalakrishnan, G., 2019, Turbomachines, Coral Publishers, Chennai, India

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